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Risk Assessments of Aging Aircraft

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Summary

The USAF believes the damage tolerance approach incorporated in ASIP process in the seventies is still the cornerstone for protecting the safety of our aging aircraft. This process is primarily deterministic in that the calculations do not quantify the reliability of the process. As indicated above, however, the reliability achieved is consistent with the new aircraft guidance identified in USAF structural specification. The USAF derives the Force Structural Maintenance Plan (FSMP) from the damage tolerance assessment (DTA). The FSMP prescribes for the maintainer how, when, and where to perform inspections to maintain safety of flight. There are cases, however, where probabilistic methods need to be used. It is the purpose of this paper to illustrate the use of probabilistic methods to ensure structural integrity.
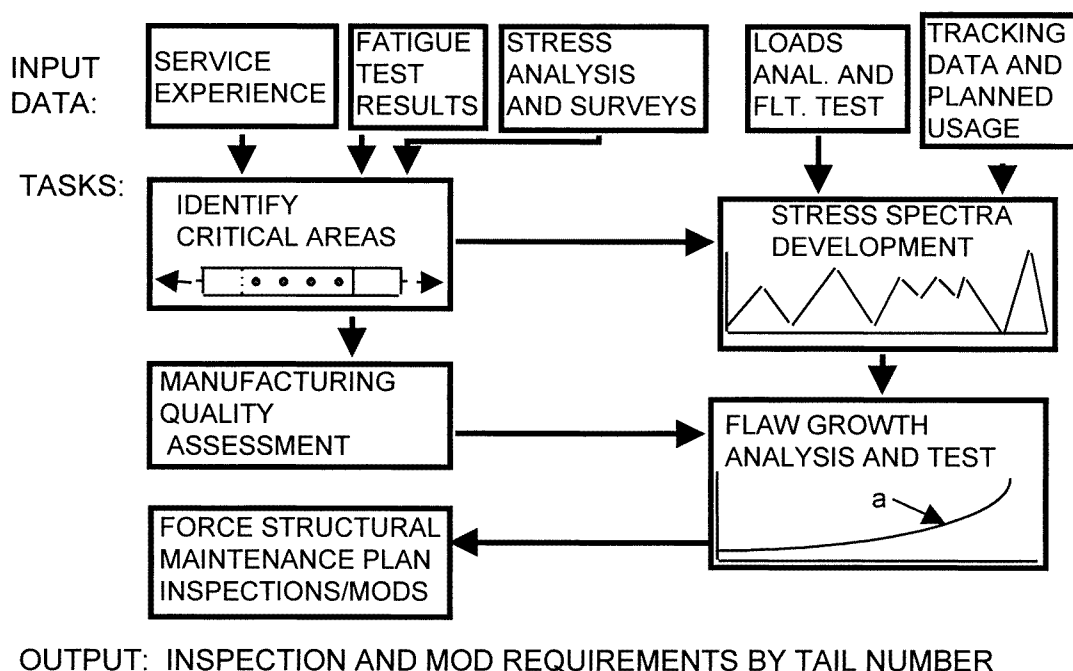
Figure 1 Damage tolerance approach

Role of Probabilistic Methods

In the seventies and eighties there was considerable activity associated with the performance of DTAs on older aircraft. The USAF sometimes found the DTA revealed critical locations that were over the safety limit. In these cases, the USAF policy is to ground (or severely restrict) these airplanes until they made an inspection. In some of these cases the inspection was so onerous they grounded the aircraft for a long time thus hindering training operations. Such a case occurred on the F-5 dorsal longeron. The inspection required approximately 1350 work hours on each aircraft. The USAF decided to modify the structure to eliminate this inspection burden. To determine the feasibility of continued use of the aircraft before the modification the USAF performed a risk assessment based on the method described in [3]. This method considers the crack length distribution and the stress distribution as random number sets. The procedure further assumes the crack growth and the residual stress functions are deterministic.

Another opportunity for a risk assessment arose when the USAF needed to keep the T-37 in operational service after the cancellation of the T-46 program. The USAF subjected the T-37 to a damage tolerance assessment. They found several areas, in particular, the wing to fuselage attachment area, where the flight hours on the aircraft exceeded the safety limit. The USAF performed an extensive risk assessment to allow these airplanes to continue in their training role until they could modify them.
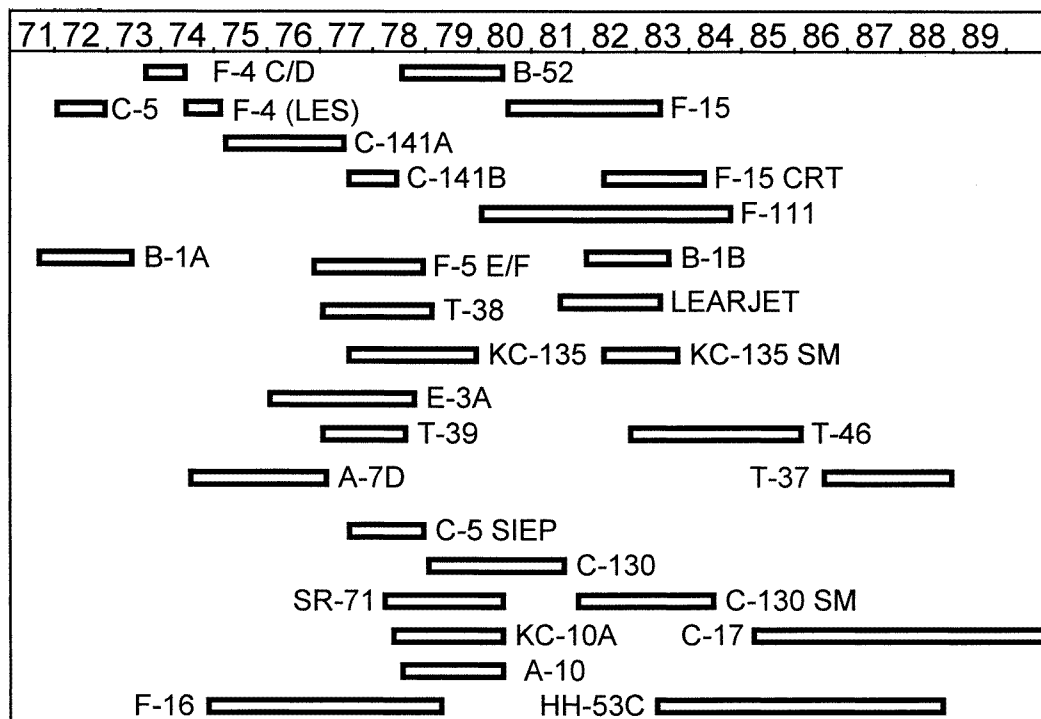
Figure 2 Damage tolerance experience - aircraft

In all of these cases such as those cited above, the risk assessment did not include the possibility of a "rogue defect" (as assumed in the damage tolerance assessment). Rather, they derived the flaw distribution from extrapolation of defects found in typical structural details. Therefore, in these cases the structural engineers made it clear to the aircraft operators they had not accounted for the rogue defect.
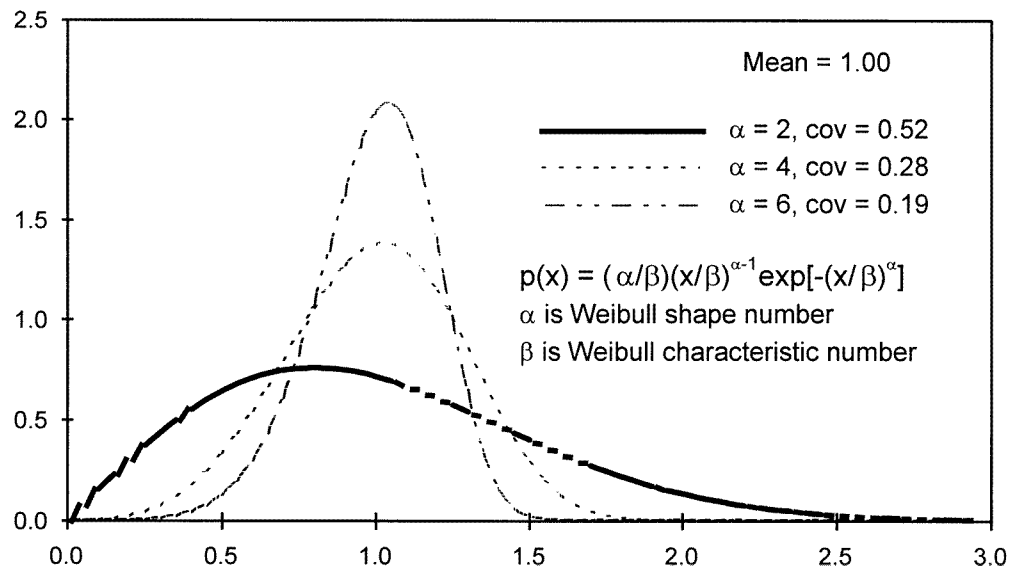
Another problem where the risk assessment is valuable is in the case where the structure is in a state of generalized cracking. In this situation the inspection intervals derived as indicated above from the deterministic DTA may be unconservative.

The USAF had an opportunity to address such a situation for the wings of the T-38 aircraft operating in the Air Training Command. In the mid-seventies, the USAF performed a damage tolerance assessment for the trainer discussed above in Air Training Command usage [3]. This study concluded they should inspect the wing center section at intervals of 1350 flight hours. They based this on an inspection capability for a corner crack of 2.54 mm and an inspection at one half of the safety limit. This was the time required to grow a crack of 2.54 mm to a critical size crack length of 5.5 mm. In the late seventies, a usage change took place that made the loading environment more severe. The USAF made a damage tolerance reassessment for this new usage. They found under the same ground rules the recurring inspection interval should be 430 hours.
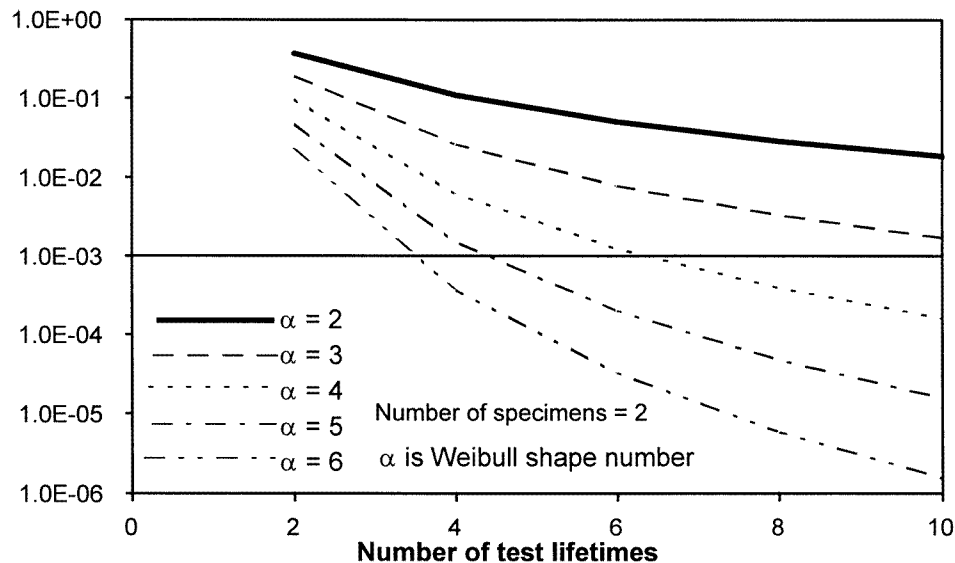
To provide an evaluation of the necessity of performing inspections at an increased rate, they performed a risk assessment for the new usage, but old inspection schedule. The assessment based on an inspection interval of 1350 flight hours showed the risk was unacceptable. When they reduced the inspection interval to 300 hours, they found an acceptably low probability of failure. Therefore, they concluded they had to improve the inspection reliability or decrease the inspection interval from that derived from the deterministic DTA.

There are other cases where probabilistic methods can complement the DTA. These cases typically involve difficulty with the performance of the DTA. One can find examples of this in the assessment of mechanical subsystems. Many of these parts are not tolerant to the size defects assumed for airframe structure. Also, the loading environments are difficult to simulate analytically. One finds another example in the high strength steel structures such as gears. In these cases some of the classical reliability approaches may be useful. As indicated in [4], W. Weibull from Sweden performed a number of fatigue experiments in the middle of the fifties. He found the results of these experiments conformed to a probability distribution, known today as the Weibull distribution.

Figure (3) shows Weibull distributions that cover the range normally found in the fatigue of aircraft structural components. One notes the coefficients of variation (the standard deviation divided by the mean) of these distributions are typically much higher than found for static strength. Figure (4) shows the reliability with 95 percent confidence as dependent on the number of test lifetimes. The results shown are for several Weibull shape numbers. This calculation assumed there were no more than two like features in the aircraft. One sees a high reliability structure is difficult to achieve when the Weibull shape number is of the order of two or three.

Figure 3 Weibull probability density functions

Mean = 1.00

α = 2, cov = 0.52
α = 4, cov = 0.28
α = 6, cov = 0.19

$$p(x) = (\alpha/\beta)(x/\beta)^{\alpha-1}\exp[-(x/\beta)^{\alpha}]$$
α is Weibull shape number
β is Weibull characteristic number

α = 2
α = 3
α = 4
α = 5
α = 6

Number of specimens = 2
α is Weibull shape number

Number of test lifetimes

Figure 4 (1 - reliability) with 95% confidence

One of the problems associated with the early applications of the safe life approach was that it did not account for the fatigue characteristics of the individual materials in the structure. Therefore, the USAF used the same scatter factor independent of the structural material or the stress spectra. The structural analyst knows today there are considerable differences between the Weibull scale numbers depending on material and spectra.

The currently acceptable structural reliability as reflected in [2] is for a single flight of an aircraft from a given population the probably of failure should be no greater than 10^{-7}. This means the desired reliability of the structure should be of the order of 0.999.

Typically, one determines the Weibull shape number through testing of multiple similar parts. An analytical example serves to illustrate how accurately one could determine the

Weibull shape number. For this purpose, one may sample a population with a known Weibull shape number. In the first case under consideration the Weibull shape number is two and the analyst selected ten random samples to simulate the testing of ten specimens. A simple transformation permits plotting these ten sample points on a graph where the Weibull distribution is a straight line. Further, on this graph the negative of the slope of that line is the Weibull shape number. Figure (5) shows the comparison of the original distribution and the sampled distribution. Figure (6) shows these distributions in the usual manner. One may use the same process to sample a distribution where the Weibull shape number is four. Figure (7) and Figure (8) show these results. One sees for small samples such as used here, the potential for error in the assessment of the Weibull shape number may be significant. In these cases the judgment on the reliability of the structural component could be in considerable error. However, if one adequately interrogates the population the results are useful. Because of the difficulties cited, the USAF recommends the application be limited to structures that are fail-safe.
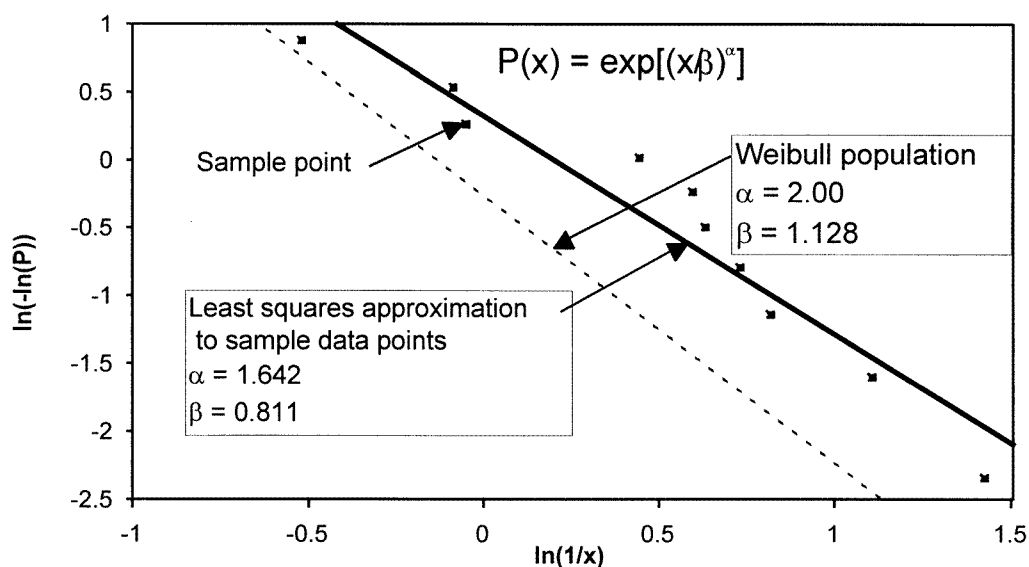
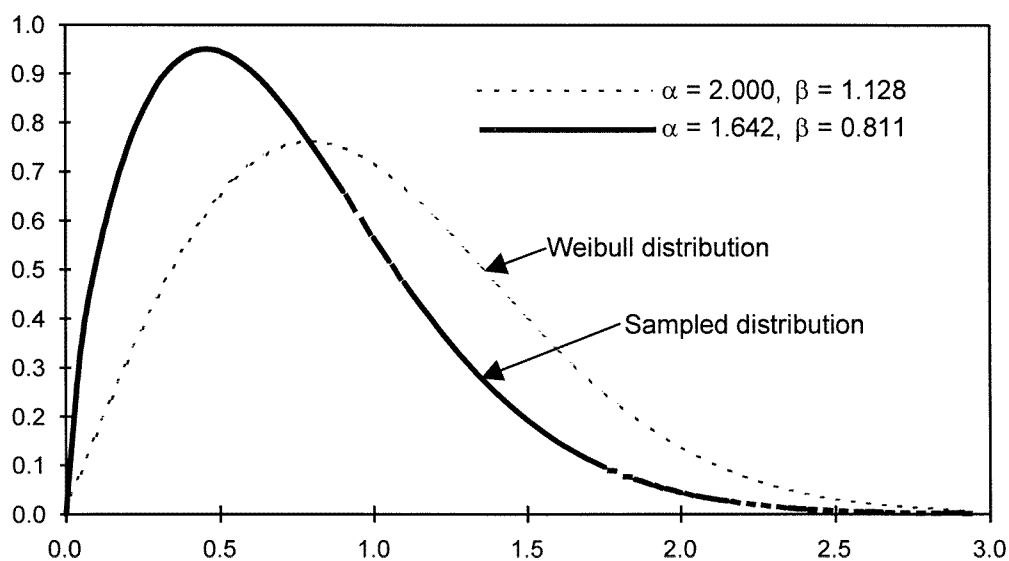Figure 5 Weibull approximation for α = 2 and 10 samples

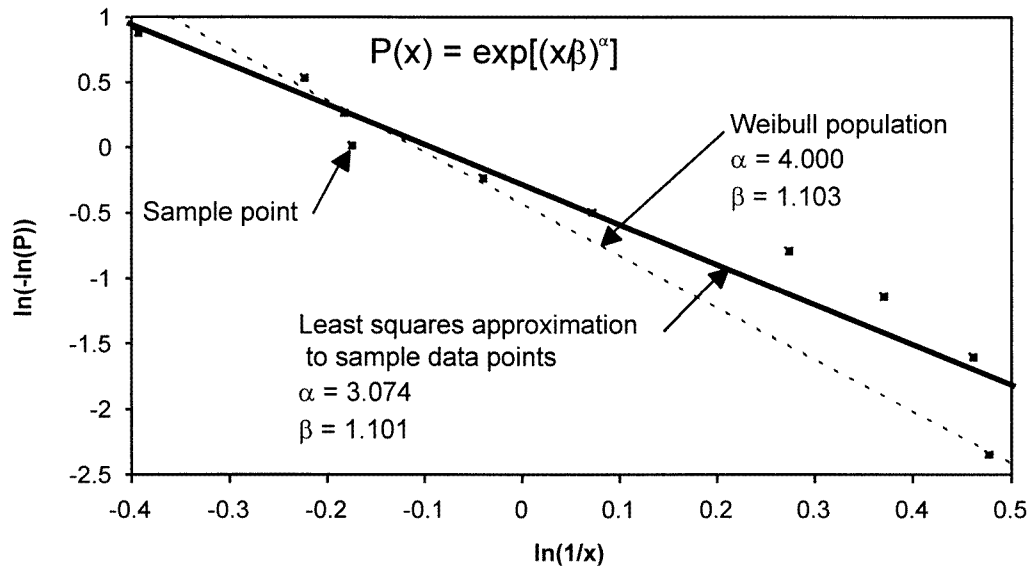Figure 6 Weibull sample comparison for α = 2
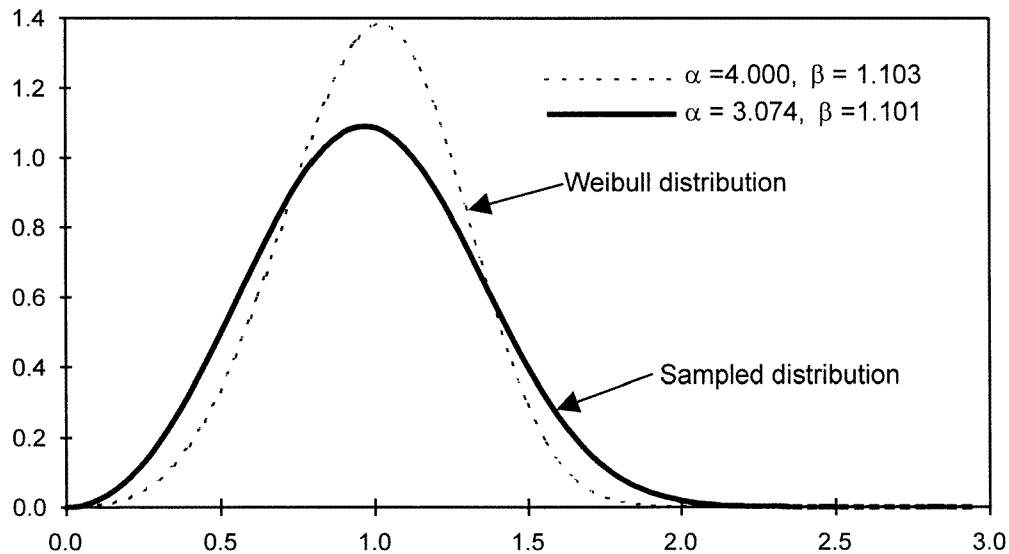
Figure 7 Weibull approximation for α = 4 and 10 samples

Figure 8 Weibull sample comparison for α = 4

Widespread Fatigue Damage

A phenomenon occurring more frequently than generalized cracking is widespread fatigue damage (WFD). WFD is a major concern in aircraft that rely on fail-safety for structural integrity. The USAF has learned WFD can degrade the fail-safe capability of a structure with cracking that is of the order of one to two millimeters [5].

A deterministic definition of WFD is the following: The onset of WFD in a structure is characterized by the simultaneous presence of cracks at multiple structural details which are of sufficient size and density whereby the structure will no longer meet its damage tolerance requirement (that is, maintaining required residual strength after partial structural failure).

In many cases this definition is difficult to apply because of the complex cracking scenarios. Further, this definition may lead to an excessively conservative determination of the time of WFD onset. An alternate definition that removes these problems is the following: The onset of widespread fatigue cracking is that point in the operational life of an aircraft when the damage tolerance or fail-safe capability of a structure has been degraded such that after partial structural failure the probability of failure of the structure falls below the thresholds specified by the procuring (or certification) agency.

For the USAF, the threshold single flight probability of failure for the intact structure is 10^{-7}. The USAF has determined the threshold for the acceptable conditional single flight probability of failure through their perception of the discrete source damage threat. In the case of the C-5A they assumed the probability of discrete source damage was 10^{-3} [6]. For the case of the 707 they assumed it was 10^{-4} [7].

One of the primary inputs to the risk assessment approach to determine the onset of the time to WFD is the distribution of cracks in the structure. The USAF has determined this distribution through teardown inspections of full-scale fatigue test articles or operational aircraft. They believe this is the best method currently available to obtain the data required to derive the probability distribution function for equivalent initial cracks in the critical areas of the structure. The word "critical" here refers to an area that could significantly contribute to the probability of failure.

The probabilistic approach also requires that the analyst determine the stress density function for each critical. The USAF derives this function from the available usage information generated by their individual aircraft tracking programs. The desired stress density function is the one for a single flight of an aircraft selected at random. The structural analyst can easily derive this function from the stress exceedance function developed as a part of the deterministic damage tolerance analysis. One can then compute the joint probability distribution of cracks and stress and integrate this function over the point set where the crack size has reached critical length. The result of this calculation is the single flight probability of failure. The time at which the probability of failure is unacceptable is the onset of WFD.

Therefore, the USAF considers the cracks in the structure and the stresses at the critical locations as random number sets. The crack growth function and the residual strength function are also treated as random functions because of the intrinsic variability of the material properties. Unfortunately, for a given population of aircraft these random number sets are not easily quantifiable. Fortunately, the variability of these functions does not appear to have a major impact on the risk of failure. Therefore, the analyst uses his best estimate of the mean of these functions in the risk assessment.

The damage scenarios in an airplane that could constitute WFD differ depending on location in the aircraft. However, typically, they fall into two categories. The first of these is multiple-site damage - characterized by cracks in multiple details in the same structural element. The second is multiple-element damage where there are cracks in multiple structural elements.

Previous efforts have shown the analyst can readily apply this type of analysis to the structures where the concern is multiple-element damage. This was the case, for example, for the KC-135 and the C-5A. The application of the risk assessment technology to the case of multiple-site damage is very much the same as it is for the case of multiple-element damage. In the case of multiple-site damage there will typically be a "boundary" that will determine if the cracking has the potential to become catastrophic. For example in the case of the fuselage lap splice, the boundary would be the crack stopper built into the structure at the frame or between the frames and its surrounding structure. This crack

arrest feature protects the integrity of the structure. The condition of the crack stopper and its surrounding structure (that is, the boundary) will determine if the damage could propagate to catastrophic failure. Therefore, the interest is primarily in the degradation of the boundary with time and not the growth of the holes in the lap splice to link-up. When one thinks of the problem in this manner, then it may be solvable in a manner similar to that used for the multiple-element damage problem. Lockheed [8] demonstrated an example this of this in their risk assessment on of the inner to outer wing joint of the C-141 aircraft.

There must be emphasis placed on the detection, through nondestructive evaluation, of cracks that could be significant for determination of the onset of WFD. As indicated above, there is a need to make an estimate of this onset based on probabilistic assessment of cracking data derived from the teardown inspection of fatigue test articles or operational aircraft. One must recognize, however, that this is only an estimate. It is not realistic to expect the analyst could determine this time with great accuracy even with the most sophisticated fracture mechanics programs. The actual time may be either somewhat earlier or later than this estimate. It is important, therefore to be able to validate this prediction with nondestructive evaluation. This is difficult because the size of defect the inspector must find is quite small. The experimental evidence to date indicates cracks of the order of two millimeters can significantly lower the fail-safe capability of certain structural configurations.

Weapon System Risk Assessments

C-5A Risk Assessment

One of the early technical challenges for this program was how long to leave this aircraft in service with the original wing design. By the mid-seventies, the USAF established the damage tolerance initial flaw size for slow crack growth structure for fastener holes as 1.27 millimeters [9]. On the basis of this flaw size the safety limit was 7,000 flight hours of the so-called 14 mission flight profiles. In this case the time for the 1.27 millimeter flaw to grow to the critical crack length was the safety limit. Since the wing was not inspectable, this was also the life limit for the wing. The USAF made a final validation of the life of the wing through a teardown inspection. They took this wing from service when it had accumulated 7,000 hours equivalent to the 14 mission flight profiles. In the teardown inspection, Lockheed examined 44,641 fastener holes in detail for cracking. They did this work in the late seventies. From the population of cracks found in this teardown inspection the USAF performed an assessment to determine the probability of catastrophic failure and the time the wing lost its fail-safety. The USAF found that at 7,000 hours the wing had initially exceeded the acceptable 10^{-7} single flight failure probability. Further, they found the wing had lost fail-safety based on a conditional single flight failure probability of 10^{-4}. This effort confirmed the USAF should take this structure out of service no later than 7,000 flight hours of equivalent 14 mission profile usage. They decided to allow the aircraft to fly to 7,000 hours with fail-safety compromised at 4,500 hours. The replacement wing box will easily meet the original life requirement of 30,000 flight hours.

C-141 Risk Assessment

The USAF found a major WFD problem in the wing at Wing Station 405 joint [8]. The USAF observed first cracking on an operational aircraft in late 1984. In early 1989, they found an aircraft with a severed beam (or spar) cap. The USAF recommended that Lockheed perform a risk assessment based on operational aircraft cracking data to assess the likelihood of catastrophic failure of the aircraft. The risk assessment, as expected, indicated the joint was extremely critical.. The USAF had found numerous cracks in the

area of the rear beam on many airplanes. In addition, they found a number of spar cap failures. Also, there have been multiple cracks discovered in the area of the forward beam on many airplanes. The risk assessment performed by Lockheed showed although inspections were somewhat effective in reducing the risk, the best alternative was to perform a modification on the joint. The USAF initiated aircraft restrictions, an inspection program, and an accelerated modification program to alleviate this problem. The action to remove WFD by a modification is similar to the earlier actions taken on the KC-135 and C-5A [6]. In the case of the KC-135 and C-5A the emphasis was on the elimination of the WFD problem rather than trying to manage it through an inspection program.

The USAF found another major WFD problem in wing lower surface fuel transfer holes (weep holes). There are more than 1500 such weep holes in each wing (both sides). The cracking experiences with the weep holes dates back to the original fatigue test. After 90,000 hours of block testing on the test article, Lockheed found cracking in many of the weep holes. Lockheed cold expanded these holes before they resumed testing with flight-by-flight loading. The additional 28,468 hours of testing showed the cold expansion was effective in controlling the weep hole cracking. Lockheed made a recommendation to WR-ALC in September of 1983 to perform the cold expansion on C-141 aircraft with 30,000 hours. In January 1993 the USAF Scientific Advisory Board (SAB) reviewed the potential for a service life extension of the C-141. They found the USAF had cold expanded weep holes on only six operational aircraft. They also found the weep hole inspection results were difficult to understand. One aircraft the USAF had found ninety-nine weep hole cracks, the longest of which was approximately 12 millimeters. They found other aircraft relatively free of cracks. However, there had been several cases where the weep hole cracks had progressed through the skin and had caused in-flight evident fuel loss. To understand this apparent anomaly, the SAB recommended a teardown inspection of an aircraft. The USAF tore down aircraft number 66-0186, in which the USAF had found ninety-nine cracks. It had 23,824 flight hours of relatively high damage usage, which converted to 44,539 damage hours (that is, hours of equivalent SLA-IIB spectrum usage) on the lower inner wing skin. The teardown inspection on aircraft number 66-0186 has revealed numerous holes with poor quality and a total of 255 cracked holes. Subsequently, WR-ALC performed an additional inspection and a limited teardown inspection on 66-9410, which had 45,202 equivalent damage hours on the inner wing lower surface. The results of the additional inspection have shown there was extensive cracking in the weep holes of this aircraft. Consequently, the USAF concluded the cracking observed in these two is representative of the aircraft with that number of equivalent damage hours. They concluded the early inspection results were unreliable. They changed the inspection procedure and validated it on a teardown inspection aircraft. The size of the cracks found led them to the conclusion there was severely degraded fail-safe capability in the wing. Also as indicated by the distribution of cracks, the cracks tend to line up which contributes to the loss of fail-safety. These airplanes were in a state of WFD. Therefore, the USAF placed the airplanes on restrictions and an inspection program. They developed an inspection program designed to preclude the cracks from reaching critical length and failing a wing panel. In addition, they developed a modification program to eliminate this problem. The modification program consisted of three parts. They found they could remove, or nearly remove, most of the cracks by reaming them. They elected to cold expand these holes. In many airplanes there were only approximately ten locations where cold expansion was not an alternative because the cracks were too large. Fortunately, at this time, the Wright Laboratory was completing a major program that would give the USAF the technology for patching metallic structures with composites. This appeared to be a more attractive alternative than the conventional metallic patches that required additional fastener holes in the lower surface of the wing. Therefore, the modification for those airplanes with a small number of large cracks would be composite patching. For aircraft that had a large number of large cracks the only

alternative was replacement of wing panels. Lockheed performed a risk assessment to better understand the severity of the weep hole cracking problem. After reviewing the results of this assessment, the USAF made recommendations for subsequent actions. They decided not to fly any aircraft that had in excess of 40,000 damage hours on the lower inner wing surface until they performed a weep hole inspection. They would inspect the remainder of the aircraft and modify them based on a one year schedule. They found weep hole cracking in practically all of the aircraft. The nondestructive inspection program revealed a total of 11,000 cracks in the weep holes in the entire fleet. The USAF found no cracks in the weep holes that had been cold expanded. WR-ALC, with the support of the Wright Laboratory [9], accomplished the tremendous task of restoring these airplanes to flight status. They repaired the wings carefully with composite patching to ensure they had not degraded structural integrity of the aircraft. They returned these airplanes to unrestricted usage when they placed them back into service.

The USAF believed that WFD of the inner wing spanwise splices was a significant factor in the C-141 continued airworthiness. They had learned this from the loss of fail-safety in the C-5A wing. In 1990 the USAF [11] estimated they could expect WFD in the spanwise splices in inner wing lower skin at about 45,000 SLA-II equivalent flight hours. They based this estimate the teardown inspection of the C-141 fatigue test article (Specimen A).. The size of cracks that could cause loss of fail-safety in the C-141 inner lower wing is in the order of 1.5 millimeters. Lockheed performed an additional assessment of the risk based on teardown inspections of wing panels taken from operational aircraft. They found significant degradation of fail-safety at 37,000 hours. The USAF made the decision to manage the safety of those airplanes above 37,000 hours by slow crack growth. This decision resulted in a very difficult inspection program [7].

707 Risk Assessment

The USAF elected in the eighties to use the 707 aircraft for Joint Stars (Joint Surveillance Target Attack Radar System). When Northrop Grumman, the contractor for Joint Stars, selected the aircraft, the configuration was the primary concern - not the age. Many of the airplanes selected were close to (or above) the original life goals of sixty thousand flight hours and twenty thousand flights established by Boeing for the 707.

The largest concern about the structure of this aircraft was the potential for the degradation of fail-safety because of WFD in the wing. Boeing performed a teardown inspection on a relatively high time aircraft in the mid seventies. The inspection performed by Boeing, completed in 1976, revealed numerous cracks in the aircraft. The cracks that caused the most concern were in the lower wing splicing stringers and the large stringers around the lower wing inspection holes adjacent to the splicing stringers. Boeing published several Service Bulletins as a result of these wing crack findings. These Service Bulletins called for either a high frequency eddy current inspection inside of the wing or an external low frequency eddy current inspection. These inspections have revealed major damage including a severed stringer and skin cracks in excess of 44 millimeters. The Boeing database, however, was not definitive enough to be usable in an assessment of the risk of failure. Consequently, the USAF contracted with Boeing to examine higher time aircraft parts taken from retired aircraft at Davis Monthan Air Force Base to quantify the risk associated with WFD [7].

Boeing performed a teardown inspection on a 707-300 wing from an aircraft at Davis Monthan Air Force Base. This aircraft, representative of the Joint Stars aircraft, had experienced 57,382 flight hours and 22,533 flights. They performed the teardown inspection on the wing lower surface and the wing stringers. Stringers and skins where Boeing used steel fasteners contained most of the cracks found. This was typically in the area of the wing skin splices and the large adjacent stringers. The beneficial effects of

the aluminum rivets attaching the other stringers to the wing skins apparently reduced the amount of cracking there. There was, however, some cracking found in these locations.

Boeing found that cracking in the aircraft in the area of the steel fasteners was quite extensive. They found a total 1915 cracks found in five sections removed from the aircraft. Most of the cracks found were small. However, they found a significant population cracked to the point of considerable concern. They found that increasing the size of the holes in the splicing stringer and the large adjacent stringer would not remove all of the cracks. About twenty percent of these holes would still have stringer cracks. Further, they found significant cracking outboard of the Wing Station 360 production joint. Therefore, the problem involved most of the wing. Typically, the large adjacent stringers had more large cracks than the splicing stringers. The largest crack found in a stringer was approximately 38 millimeters in length. It was near the point of rapid fracture. There were, however, many cracks found that would have gone to failure in the planned life span of the Joint Stars aircraft. There was a concern about cracking that would degrade the capability of the structure to sustain discrete source damage. There was also a concern about the fatigue failure of the stringers and subsequent catastrophic loss of the aircraft after a skin failure.

Boeing calculated the stress intensity of each of the cracks found. They then determined for each of them the size of the corner crack with the same stress intensity. From these cracks, the USAF derived the crack distribution function. They used a population taken from the largest of them to approximate the crack distribution with a two parameter Weibull distribution function. It is typical that a single Weibull distribution function will not approximate the longer cracks as well as the shorter cracks. This is not a problem since only the longer cracks will have a significant effect on the risk of failure.

The USAF needed two stress distribution functions for the assessment. The first is the stress distribution function for the intact structure. Boeing derived this in the usual manner from the intended usage of the aircraft, the external load analysis, and the stress analysis of the wing. Second, for the cases where discrete source damage was present they determined the local stress increase from the damage. In many cases the local stress increased to the point where there was significant plastic deformation of the structure. When this occurs it is essential the plastic deformation be included in the analysis. A linear analysis in these cases would likely lead to serious errors in the determination of risk.

For the cases of discrete source damage the maximum single flight failure probability allowed was 10^{-3} and for the intact structure case the maximum single flight failure probability allowed was 10^{-7}. For the stated criteria for discrete source damage, the USAF found significant degradation of fail-safety beyond 50,000 flight hours of commercial usage. Therefore, for some aircraft, there will be unacceptable fail-safety degradation before the end of the planned 20,000 hours of Joint Stars usage. This will occur for Joint Stars aircraft with more than 36,000 commercial usage hours. Further, for the case of no discrete source damage, there will be safety degradation beyond 58,000 hours of commercial usage. Therefore, aircraft with initially more than 44,000 hours of commercial usage will have a high probability of failure before operationally flying 20,000 hours.

There are two possible approaches for solution of this problem. The first is to remove the steel fasteners in the area of concern in the lower wing surface and perform an eddy current inspection. If the inspector finds no indication of a crack or if increasing the size of the hole would remove the indication, then this hole would be cold expanded. For cracks that are too large for this remedy, the USAF could utilize a repair such as composite patching. This approach appears to be viable for aircraft with less than 45,000

commercial usage flight hours. It also may be viable for aircraft in the 45,000 to 55,000 flight hour range. A second alternative would be to replace the wing panels and stringers in the area of concern. This may be the only alternative for aircraft with more than 55,000 commercial usage flight hours.

Widespread Fatigue Damage Example Risk Assessment

The following example illustrates some of the essential features of the risk analysis process. The example determines the risk of catastrophic failure for both the intact and partially failed structure of a hypothetical aircraft designed for a 30,000 hour life. The aircraft is to fly only one mission that is two hours in length. The aircraft has one critical area with 500 fastener holes. The initial crack distribution is the crack distribution function derived from a teardown inspection. Figure (9) shows the corresponding crack density function. Figure (10) shows the corresponding crack distribution function. For the intact structure, Figure (11) shows the stress exceedance function for each of these holes. Figure (12) shows the corresponding stress probability distribution function, derived from the exceedance function. Figure (13) shows the stress density function. The threat of discrete source damage is 10^{-3}. For the partially failed structure, only ten of the 500 fastener holes have their stress increased to 1.5 times the stress for the intact structure. Figure (14) shows the residual stress function. The crack growth function modifies the initial crack distribution function so the crack probability distribution has the correct time dependence. Figure (15) shows the crack growth function. Figure (16) shows the final function needed for the calculation of risk. This is the inspection probability of detection function.
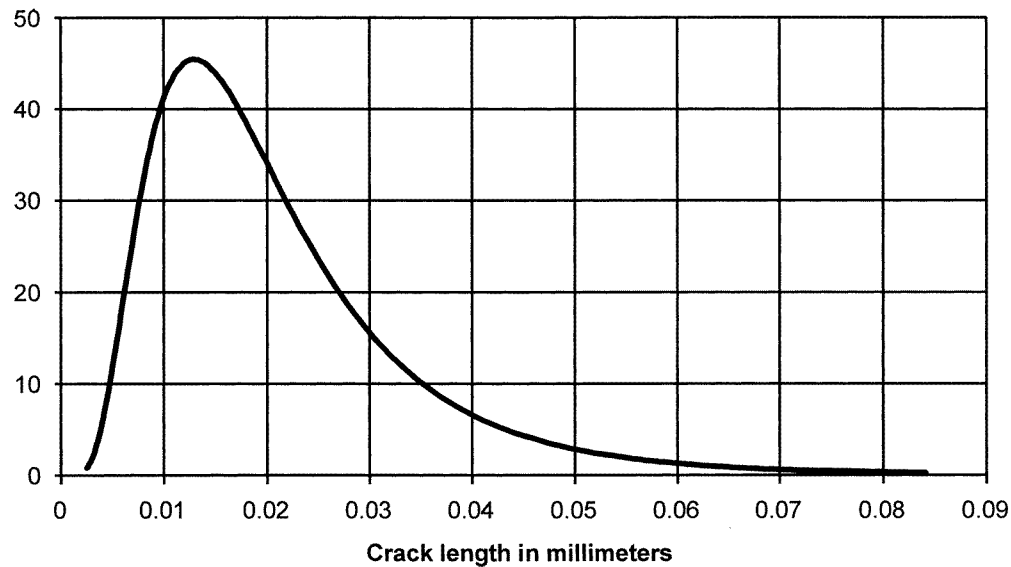

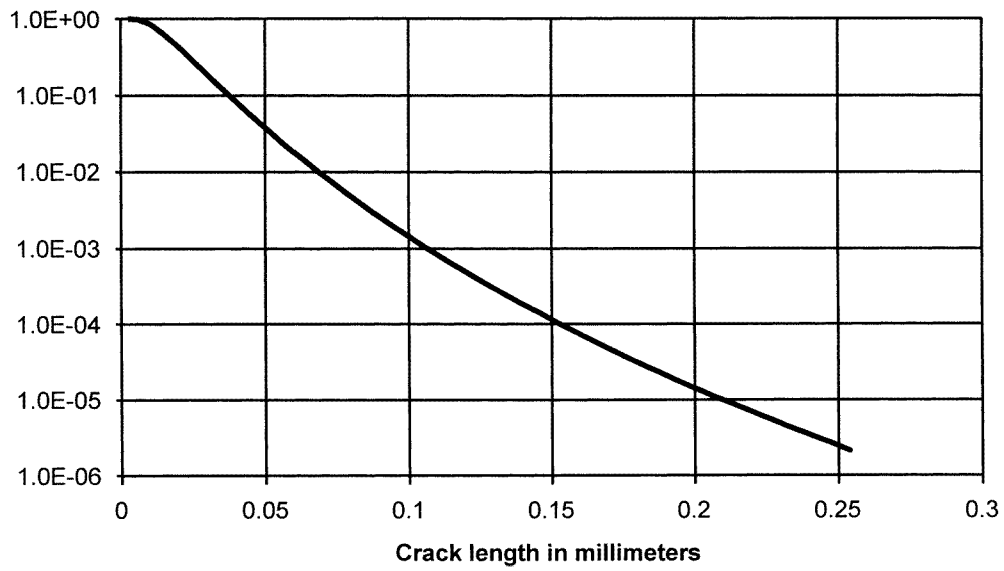Figure 9 Crack density function from the A-7D

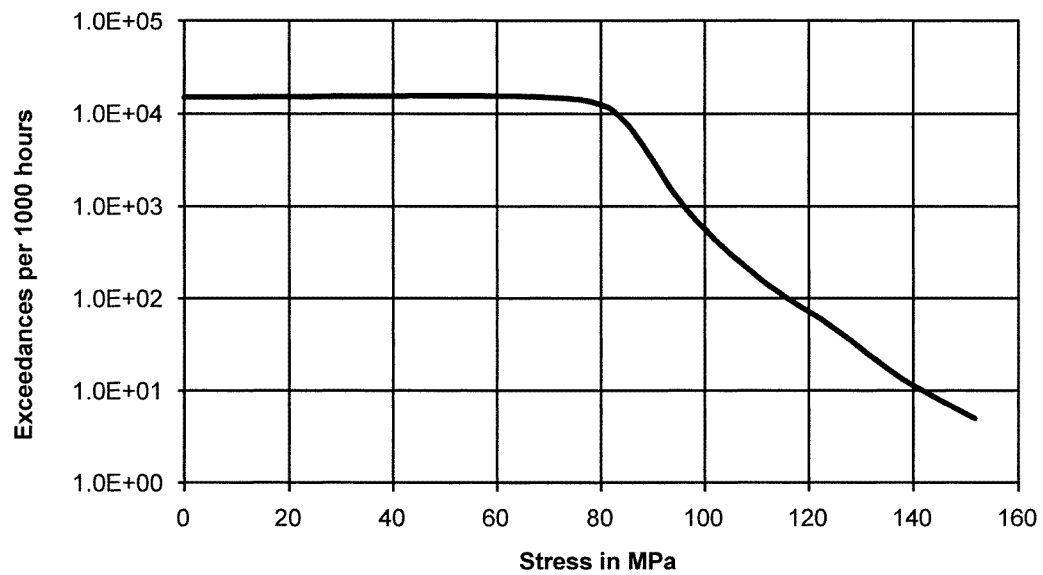**Figure 10 Crack distribution function from the A-7D**

Figure 11 Stress exceedance function
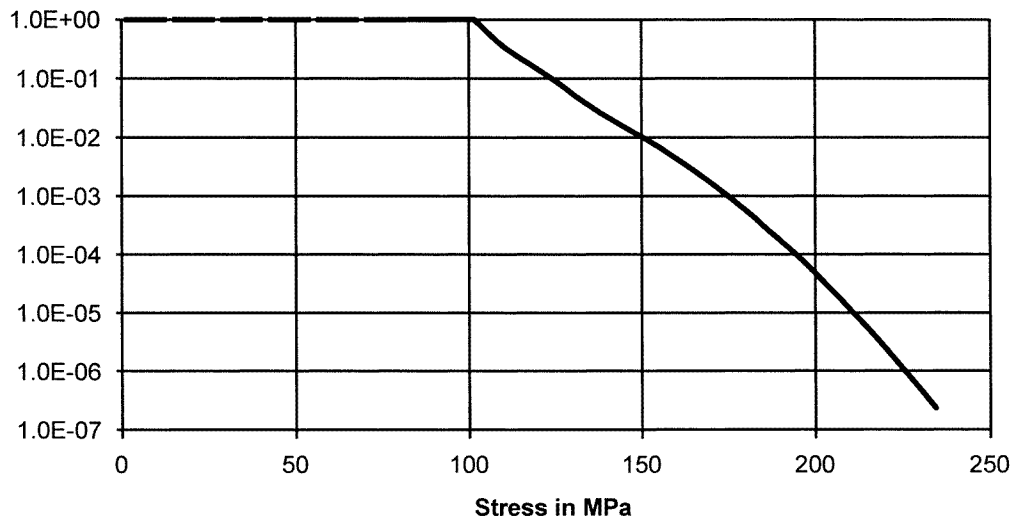
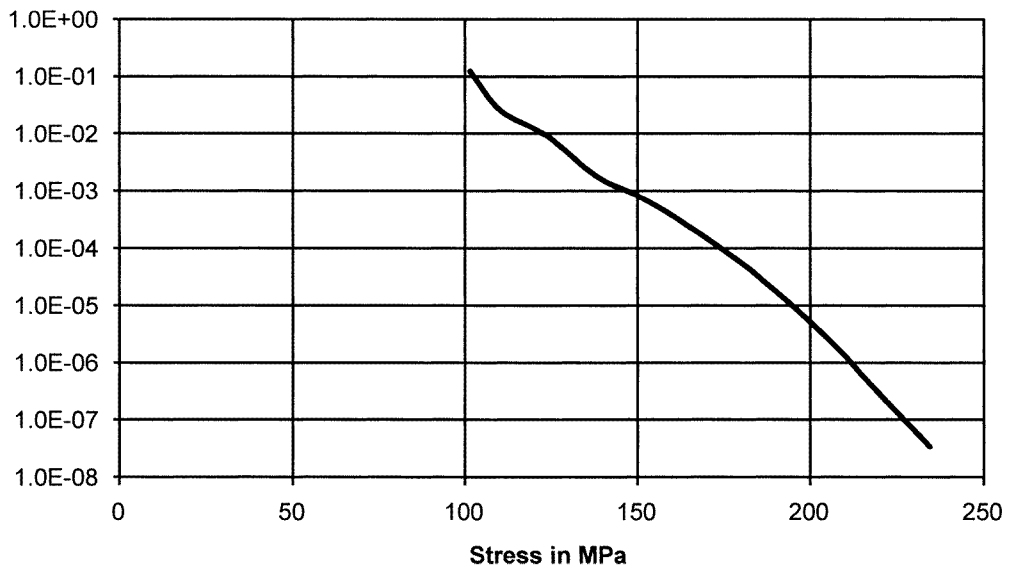**Figure 12 Stress probability distribution function**

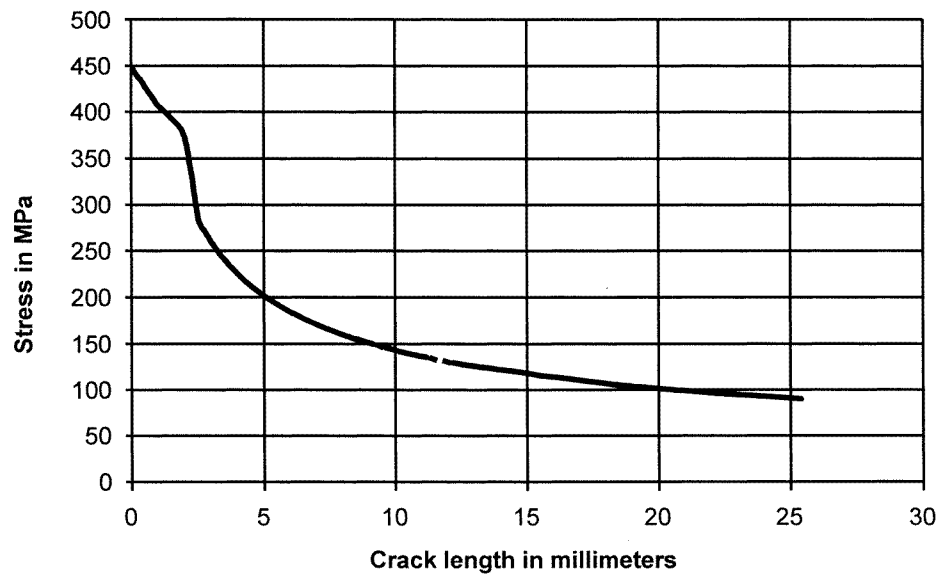**Figure 13 Stress probability density function**
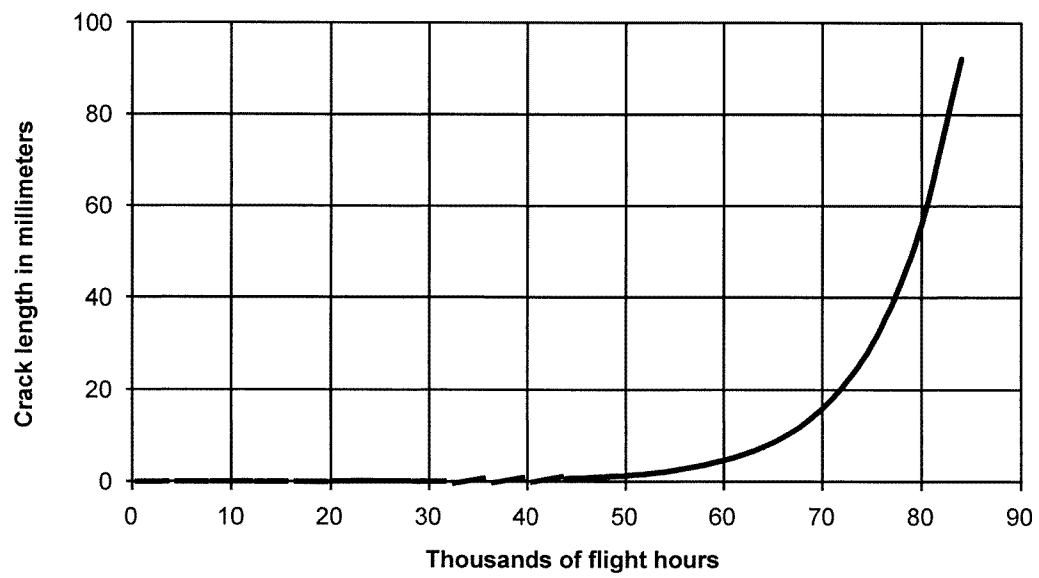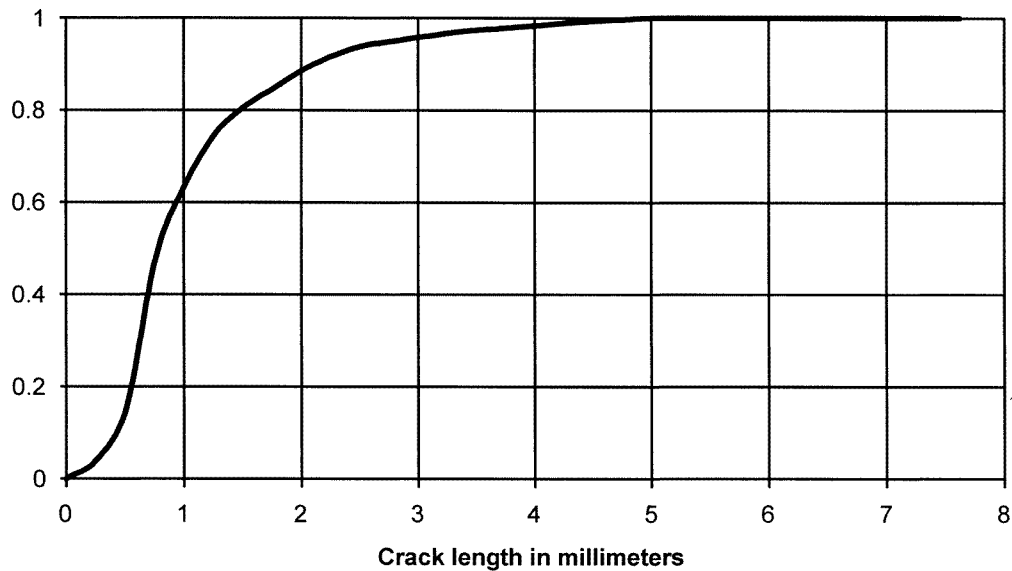
Figure 14 Residual stress function

Figure 15 Crack growth function

Figure 16 Probability of detection function

Figure (17) shows the single flight probability of failure for the intact structure without inspections. From this figure, one sees the risk exceeds the 10^{-7} threshold of acceptability at about 22,000 flight hours. From Figures (14), (15), and (16) the analyst can determine the damage tolerance inspections. The first inspection is at 7600 flight hours and the inspection interval following the first inspection is 5000 hours. Figure (18) shows the single flight probability of failure for the intact structure with inspections. One sees these inspections are quite effective in reducing the risk of failure and containing the risk within acceptable limits to 30,000 flight hours. It is clear from this figure that on the basis of the inspection capability assumed and the inspection interval derived from the damage tolerance methodology the risk is increasing significantly. Therefore, one must make a reduction in the inspection period if one intends to fly the aircraft significantly beyond its original life of 30,000 flight hours.

Single flight failure probability
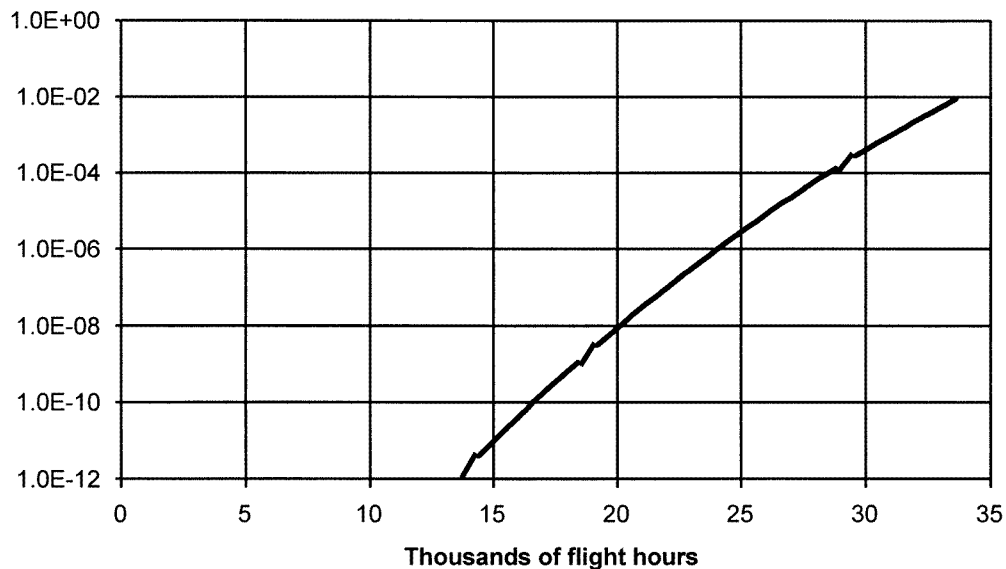
Figure 17 Intact structure with no inspections

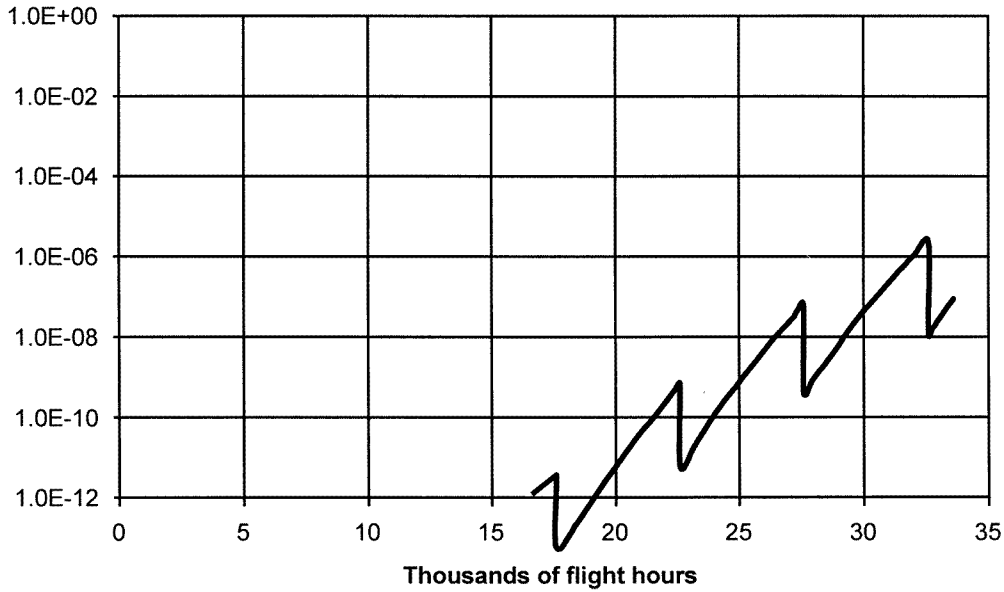Single flight failure probability

Figure 18 Intact structure with inspections

Figure (19) shows the single flight failure probability for the partially failed structure without the effect of inspections. This is the conditional probability for the structure damaged from an external source. One sees the risk crosses the threshold of acceptability for this case (that is, 10^{-4}) at approximately 16,000 flight hours. The aircraft has degraded fail-safe capability long before the time the intact structure has reached the unacceptable risk threshold. Figure (20) shows the influence of inspections on the probability of failure. One sees the inspections are essentially ineffective in reducing the risk for this case. This example clearly illustrates the damage tolerance derived inspection program may not adequately protect the fail-safety of an aircraft in the presence of widespread fatigue cracking.
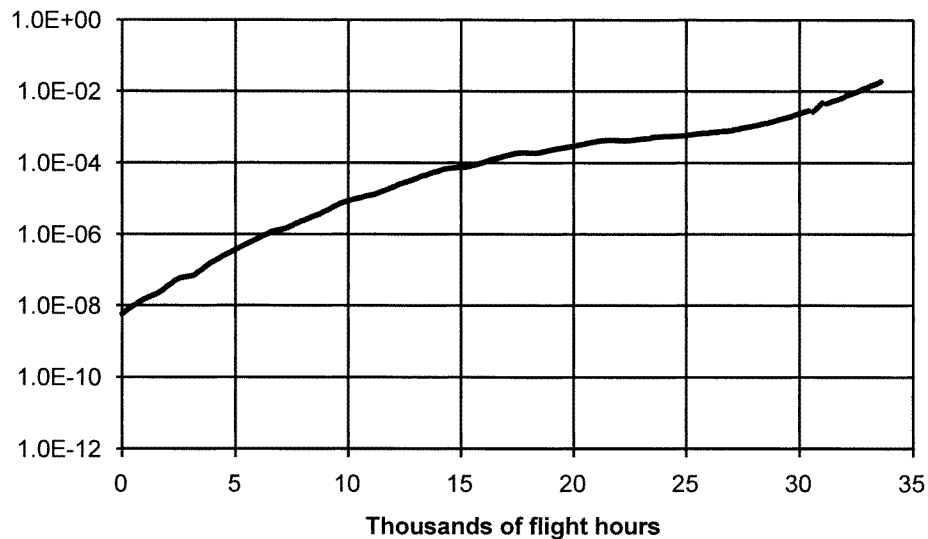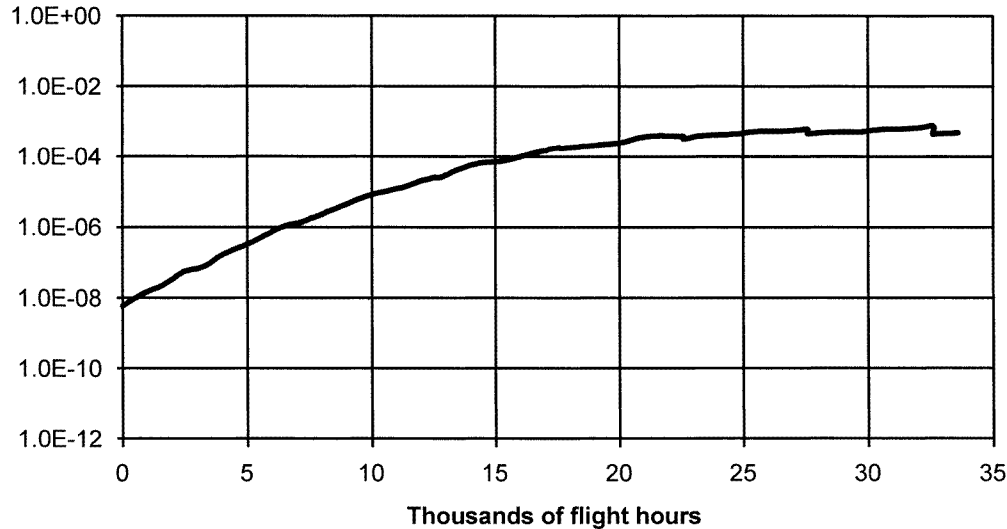
Single flight failure probability

Figure 19 Damaged structure with no inspections

Single flight failure probability

Figure 20 Damaged structure with inspections

Conclusions

As indicated above, the cornerstone for protecting the safety of USAF aircraft is damage tolerance. There are some cases, however, where probabilistic methods find an important use. One approach that appears to be attractive especially for mechanical subsystems is the use of reliability analyses based on testing. In some cases these methods can provide satisfactory solutions where a damage tolerance assessment may be impractical. The USAF believes the process may apply to mechanical subsystems since they are typically fail-safe by design.

A major problem in aging aircraft is WFD. It is essential to establish an estimate of the time of onset of this problem. The USAF does this through the analysis of data derived from teardown inspections of fatigue test articles and of operational aircraft. They will need to corroborate these estimates through the use of detailed inspections of suspect structural elements. In some cases the nondestructive inspection capability does not exist to economically find WFD size cracks. The USAF must continue their effort to attain this capability. Once the aircraft operator determines the aircraft has reached the time of onset of WFD, he needs to make modifications of the structure to remove this problem.

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